SCHLIEREN OBSERVATION OF SUPERSONIC DISCHARGE

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COPY FOR HEAD OF POSTGRADUATE SCHOOL

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September 24, 1946

Captain W. H. Buracker
Room 5-233
Massachusetts Institute of Technology
Cambridge 39, Massachusetts

Thesis Work of LT E. L. PERRY, USCG
LT L. W. A. RENSHAW, USCG
LCDR W. W. SIMONS, USN
LCDR J. S. BOWEN, USN

Dear Captain Buracker:

The thesis by Lieutenants E. L. Perry and L. W. A. Renshaw entitled "Schlieren Observation of Supersonic Discharge" presents pressure measurements and Schlieren photographs of supersonic streams discharging into an exhaust space under various conditions. The photographs show interesting detail which in general corresponds to analytical results. The most significant observation was a comparison of two supersonic streams alike in average conditions but differing in thickness of the boundary layer. The effect of boundary-layer thickness on the nature of the shock pattern is shown clearly.

The thesis by Lt. Comdrs' W. W. Simons and J. S. Bowen entitled "Investigation of the Condensation Shock in Air by Use of the Schlieren Method" presents pressure measurements and Schlieren photographs of the shock patterns when water vapor in air condenses to form a fog of liquid or solid particles. It has extended our knowledge of the conditions which control condensation and of the condensation shock which accompanies it.

From either of these theses a paper could be prepared which would be published in one of the journals of the professional societies.

Yours truly,

/s/ Joseph H. Keenan

Joseph H. Keenan

SCHLIEREN OBSERVATION

OF

SUPERSONIC DISCHARGE

BY

Lieut. E. L. Perry
B.S., U.S.C.G. Academy
1941

and

Lieut L.W.A. Renshaw
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1941

Submitted in partial fulfillment of the

requirements for the degree of

Master of Science

at the

Massachusetts Institute of Technology

1946

MASSACHUSETTS INSTITUTE OF TECHNOLOGY
77 Massachusetts Avenue
Cambridge, Massachusetts

September 15, 1946

Professor Joseph S. Newell
Secretary of the Faculty
Massachusetts Institute of Technology
77 Massachusetts Avenue
Cambridge, Massachusetts

Dear Professor Newell:

Herewith we submit our thesis entitled "Schlieren Observation of Supersonic Discharge" in partial fulfillment of the requirements for the Degree of Master of Science in Naval Construction and Engineering at the Massachusetts Institute of Technology.

ACKNOWLEDGEMENT

With pleasure we acknowledge the help given us by Professor Joseph H. Keenan and Professor E. P. Neumann of the Mechanical Engineering Department. Thanks are also due to Dr. Joseph Kaye and Mr. Ferdinand Lustwerk. Mr. Lustwerk rendered invaluable assistance in developing laboratory equipment and technique.

Professor Joseph H. Keenan suggested the thesis topic.

TABLE OF CONTENTS

SUMMARY

This work was undertaken to observe the effect on the discharge phenomenon of a supersonic air stream due to a change in Mach Number and a change of boundary layer thickness at constant Mach Number. Two (2) two-dimensional nozzles were designed using the Prandtl Theory, one having a Mach Number of 1.85 and the other a Mach Number of 1.39. A third nozzle was formed by adding a lenght of straight tube to the profile of the first nozzle to bring the Mach Number down to 1.39 by friction. All nozzles were designed for the same flow per unit area in the exit.

A comparison of the discharge of the first and second nozzles should show the effect of Mach Number, whereas a comparison of the second and third nozzles should show the effect of boundary layer thickness. The comparisons were made by Schlieren photographs and pressure measurements by mercury manometers at a point one eighth (1/8) inch from the exit of the nozzle and in the discharge chamber. It is noted that the nozzles were mounted perpendicular to the knife edge in the apparatus.

The results of the first comparison are not too conclusive. Further study in this line is recommended. The second comparison shows that a thick boundary layer cannot support anything resembling a transverse shock whereas a thin boundary layer will. Pressure measurement revealed that even in the thinest boundary layers we were able to obtain there was no abrupt rise in pressure in the exit of the nozzles - like that expected in frictionless flow - as the exhaust pressure was increased. It is pointed out that the pressure was measured at the wall at a point one eighth (1/8) inch from exit. The photographs show that as the exhaust pressure is increased, the

oblique shock tends to creep back from the exit. This is shown in Figures VIII, IX and X. The gradual rise in exit pressure shown by our measurements may be due to this creeping back of the oblique shock over the pressure tap. Figure I shows that there were slight discontiuities in the pressure curve for the high Mach Number discharge. The photographs in this region - Figures XI, XII and XIII - depict this instability in the flow.

It is recommended that further work of this nature be carried out with the nozzles mounted parallel to the knife edge of the Schlieren apparatus in order to observe more precisely the contribution of the boundary layer to the discharge phenomena.

INTRODUCTION

The academic interest in the flow of fluids at supersonic velocities has recently become of practical importance due to the development of gas turbines etc. The theory of the manner in which a supersonic stream from a nozzle or tube adjusts itself to the pressure in the exhaust space is well developed.

This work proposes to investigate and observe by Schlieren methods of photography the manner in which such adjustments are accomplished and the effects of different Mach Numbers and different boundary layer thicknesses on the phenomena.

Three (3) two-dimensional nozzles were designed. These were fitted with plane glass sides so that the flow in the exit and discharge chamber could be observed by a Schlieren apparatus. The first of these nozzles (designated Nozzle #1 and shown in Figure XXXVI, Appendix B) was designed to have as little boundary layer as possible and a Mach Number of 1.85. The second nozzle (designated Nozzle #2 and shown in Figure XXXVII, Appendix B) was designed for the same flow per unit area at the exit and a Mach Number of 1.39. A comparison of these two nozzles should show some effect of Mach Number change on the discharge. The boundary layer should be small in each since they are very short.

To compare the discharge at the same Mach Number and different boundary layer thicknesses a straight portion was added to the contour of Nozzle #1 to reduce the Mach Number by friction to the same value as that of Nozzle #2 - (1.39). It was anticipated that some adjustment of the length of the straight portion would have to be made to bring the Mach Number to 1.39. This was later found to be the case.

The laboratory procedure consisted of mounting the nozzles in the Schlieren apparatus and taking suction with a steam jet air ejector. Air at room temperature and atmospheric pressure was used as supply to all the nozzles. It is noted that in order to maintain the same flow per unit area for Nozzle #2, a specially designed reducing fitting shown in Fifure XXXIX was used to reduce the inlet pressure to two thirds (2/3) atmosphere.

Starting with the lowest pressure we could obtain in the

PROCEDURE

Three (3) two-dimensional nozzles were designed. These were fitted with plenum glass sides so that the flow in the exit and discharge region could be examined by a Schlieren apparatus. The first of these nozzles (designated Nozzle #1 and shown in Figure XXXI, Appendix B) was designed to have as little boundary layer as possible and a Mach Number of 1.85. The second nozzle (designated Nozzle #2 and shown in Figure XXXII, Appendix B) was designed for the same flow per unit area at the exit and a Mach Number of 1.75. A comparison of these two nozzles should show some effect of Mach Number change on the discharge. The boundary layer should be small in each since they are very short.

To compare the discharge at the same Mach Number and different boundary layer thicknesses a straight portion was added to the contour of Nozzle #1 to reduce the Mach Number by friction to the same value as that of Nozzle #2 - (1.75). It was calculated that some adjustment of the length of the straight portion would have to be made to bring the Mach Number to 1.75. This was later found to be the case.

The laboratory procedure consisted of mounting the nozzles in the Schlieren apparatus and taking readings with a steam jet air ejector. Air at room temperature and atmospheric pressure was used as supply to all the nozzles. It is noted that in order to maintain the same flow per unit area for Nozzle #2, a specially designed reducing fitting shown in Figure XXXIII was used to reduce the inlet pressure to ten twelfths (10/12) atmosphere.

Starting with the lowest pressure we could obtain in the

discharge the exhaust pressure was allowed to increase in steps until the pressure shocks were seen to move back into the nozzle. Readings of the exhaust pressure and the pressure one eighth (1/8) inch upstream from the exit plane were made by mercury manometer and recorded. Photographs were made at each step using the Edgerton Flash Unit described in Reference (1). Graphs of exit pressure vs exhaust pressure were plotted.

RESULTS AND DISCUSSION OF RESULTS

The results of the experiment are shown in Figures I to XXXV. A comparison of Figure I and Figure II would indicate that a phenomenon more closely approaching a theoretical transverse shock is found in flow at higher Mach Number. The break in the pressure curve for Nozzle #1 (Mach Number 1.85) at an exhaust pressure of about 240 mm Hg. is much more pronounced than for any in the curve for Nozzle #2 (Mach Number 1.39). Examination of Figures X, XI and XII shows some instability of the discharge at the instant the shock occurs at the exit of the nozzle for the higher Mach Number. No such instability was observed at the lower Mach Number (1.39). Figures XXV, XXVI and XXVII show, however, what appears to be a transverse shock at the lower Mach Number. It is believed that the comparatively smooth pressure curve for Nozzle #2 is caused by the length of the shock. Apparently the flow separates from the tube wall near the exit and the shock passes smoothly up the nozzle as the exhaust pressure increases ; whereas at the higher Mach Number the shock is much shorter and the flow less stable. We were unable to stop the shock in the exit of this nozzle.

Under all conditions the pressure in the stream adjusted itself to a lower exhaust pressure by the expansion wedges expected from the Meyer Theory of flow around a corner. This is shown in Figures IV, XXII and XXVIII. Small and moderate adjustments to a higher exhaust pressure were made in all cases by the medium of an oblique shock. There was a tendency for the oblique shock to creep back into the nozzle as the exhaust pressure increased. It is observed that this tendency became very pronounced in the case

of the thick boundary layer. It is possible that the gradual rise in the observed exit pressure as the exhaust pressure is increased is due to the oblique shock creeping back over the pressure tap which is located one eighth (1/8) inch from the exit. In that event the observed pressures are probably not the true pressures in the center of the stream at exit.

A comparison of Figure II and III show a marked similarity in the pressure relations of the two discharges at the same Mach Number (1.39) but different boundary layer thicknesses. It is noted that the curve for Nozzle #3 with a thick boundary layer is displaced to the right by about 15 mm Hg. on the exhaust pressure scale.

The mechanism by which the pressure in the stream adjusts itself to a considerably higher exhaust pressure is shown in Figures XXXII to XXXV and Figures XXIV to XXVII to be somewhat different in these two cases. In the case of the thick boundary layer Figures XXXII to XXXV show that nothing resembling a transverse shock occurs. Instead, the boundary layer, which is subsonic, appears to increase in area while the supersonic stream decreases in area; thus the pressure rises to that of the exhaust chamber. The oblique shocks which are set up and reflect downstream appear to originate at the point where contraction of the supersonic stream begins. It is possible that this apparent enlargement of the boundary layer cross section is actually a flow separation from the wall. The observation that this phenomenon occurs only in the case with thick boundary layer supports the former assumption, however.

It is recommended that further work on this point be carried

out with the nozzle mounted parallel to the knife edge of the Schlieren apparatus so that a better idea of what is going on in the boundary layer may be obtained.

In the case of the thin boundary layer no such separation or enlargement is observed. What appears to be a transverse shock with perhaps a little separation is shown in Figures XXiv to XXVII.

Investigation of the effect of Mach Number on the discharge with thick boundary layer is also recommended. It would be interesting to make observations at a Mach Number of 1.39 and with a boundary layer intermediate in thickness between the two cases used in this work.

As is noted in Appendix A the length of straight tube necessary to reduce the Mach Number of Nozzle 1 to that of Nozzle 2 was calculated to be 10.35 inches. Actual experiment revealed that this value should be 6.02 inches and the length was accordingly reduced to that value.

Due to extremely low temperatures of the stream it was practically impossible to prevent the condensation of moisture on the outside surfaces of the glass plates. This resulted in smudges similar to those shown in Figures IX, X, XXVIII and XXX.

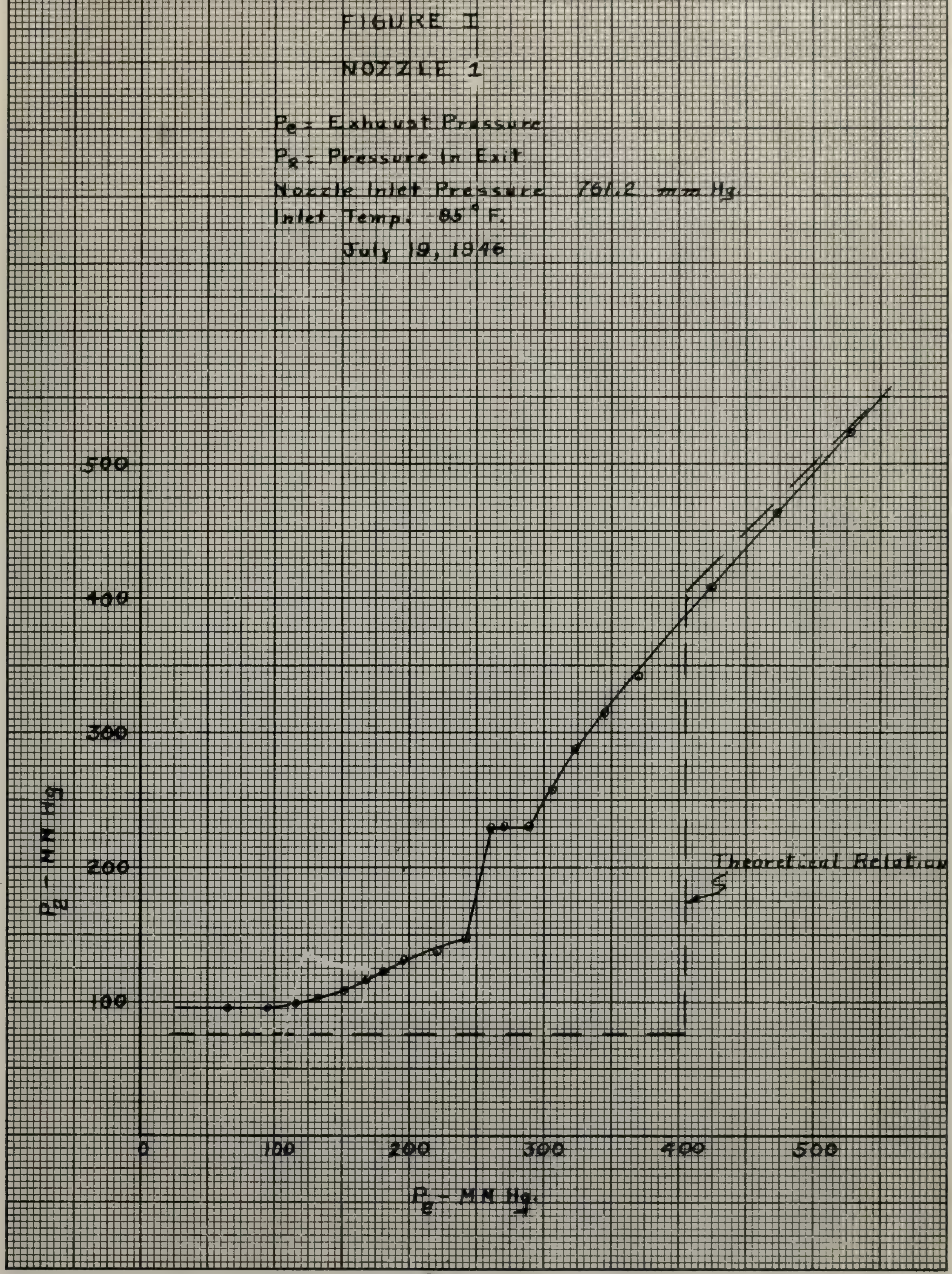


FIGURE I
NOZZLE 1
P_e = Exhaust Pressure
P_2 = Pressure in Exit
Nozzle Inlet Pressure 761.2 mm Hg.
Inlet Temp. 85° F.
July 19, 1946
Theoretical Relation
P_2 - MM Hg
P_e - MM Hg.
0
100
200
300
400
500

FIGURE II

NOZZLE 2

P_e = Exhaust Pressure

P_2 = Pressure in Exit.

Nozzle Inlet Pressure 508 mm Hg.

Inlet Temp 85°F

July 13, 1946

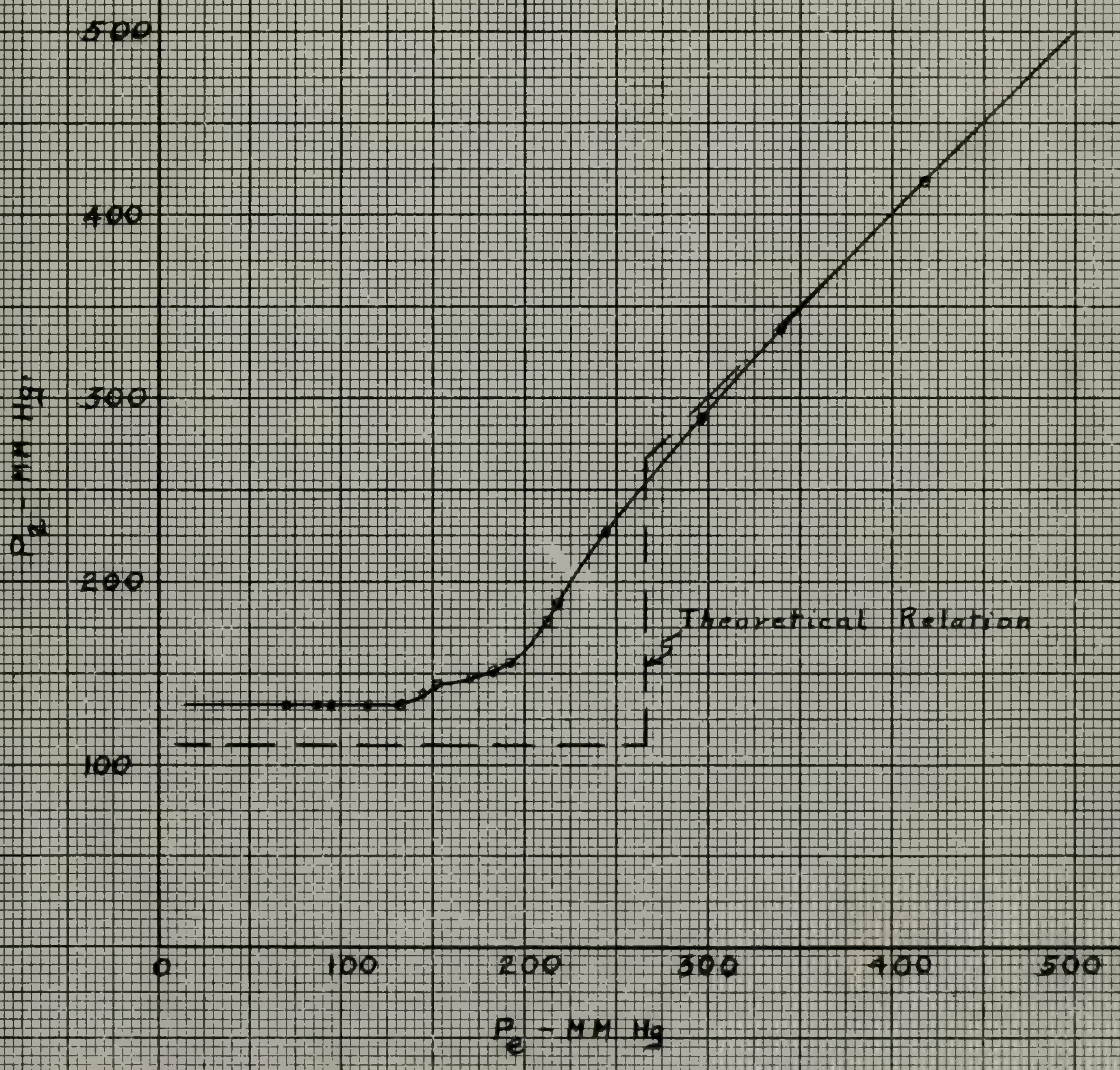


FIGURE III

NOZZLE 3

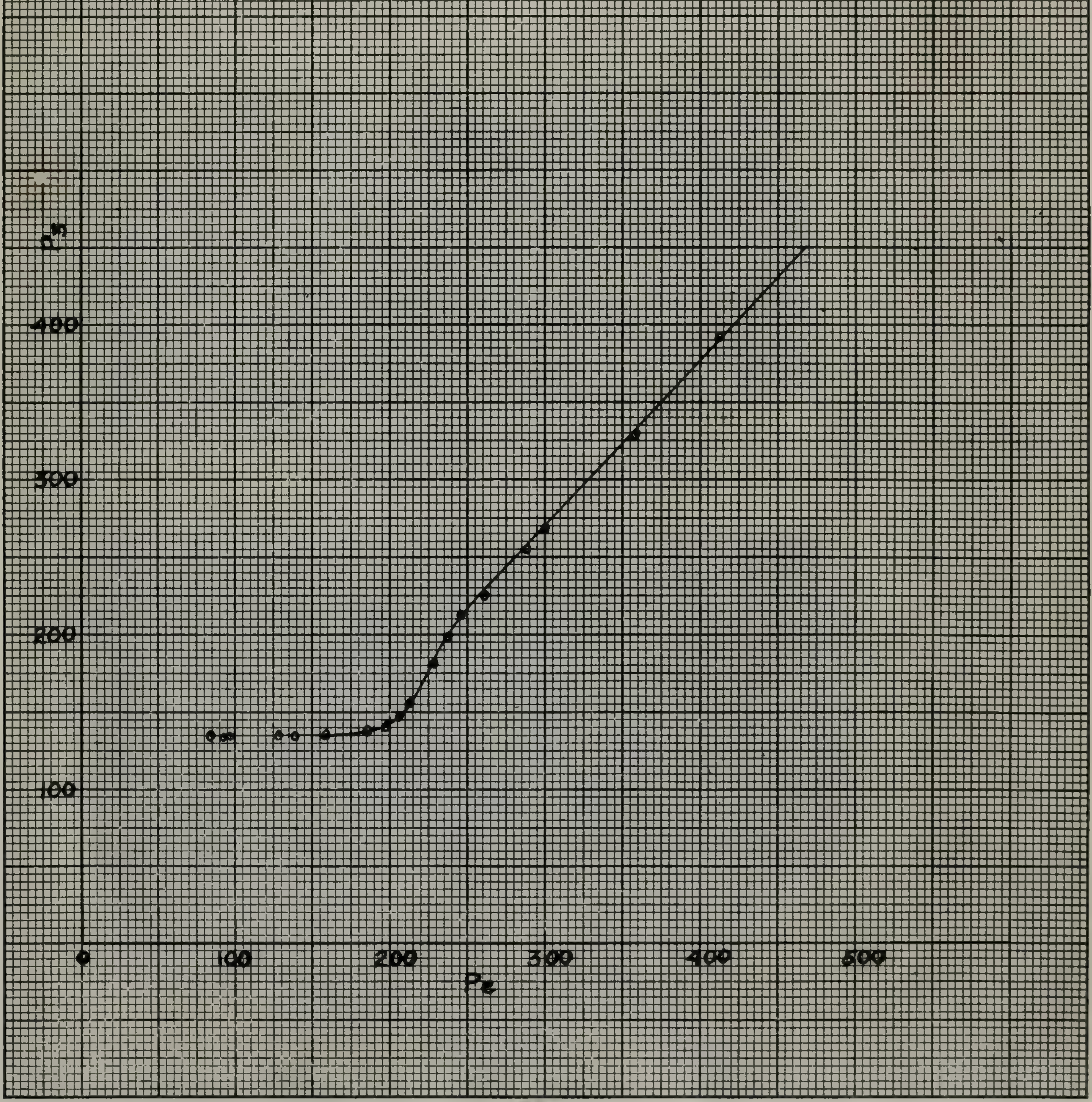
P_E = Exhaust Pressure mm Hg

P_3 = Pressure in Exit "

Nozzle Inlet Pressure 764.4 mm Hg

Tube Inlet Pressure 93.4 "

July 22, 1946



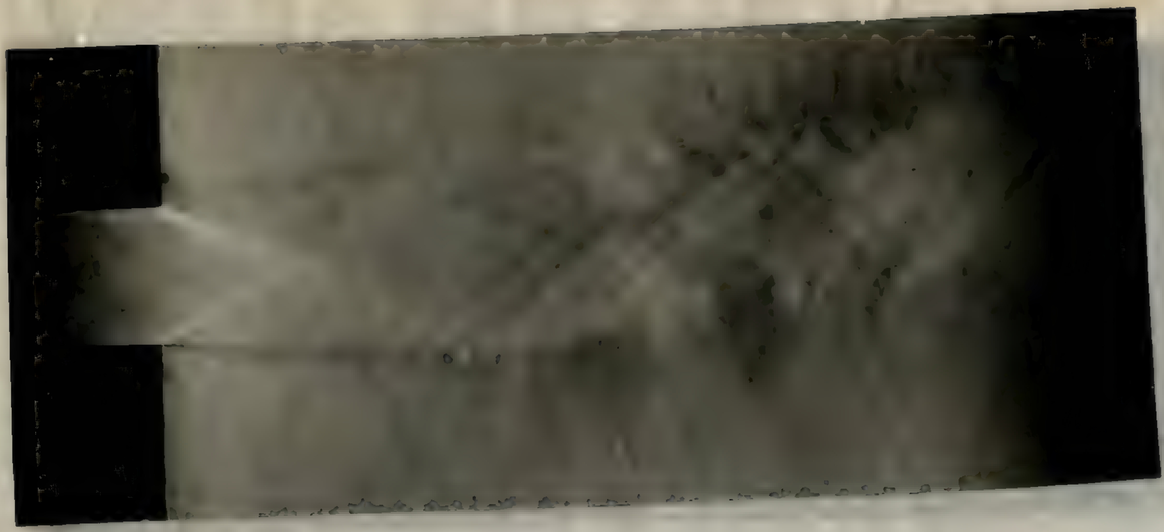


FIGURE IV Nozzle #1 $P_e = 74$ $P_2 = 95$ Flash

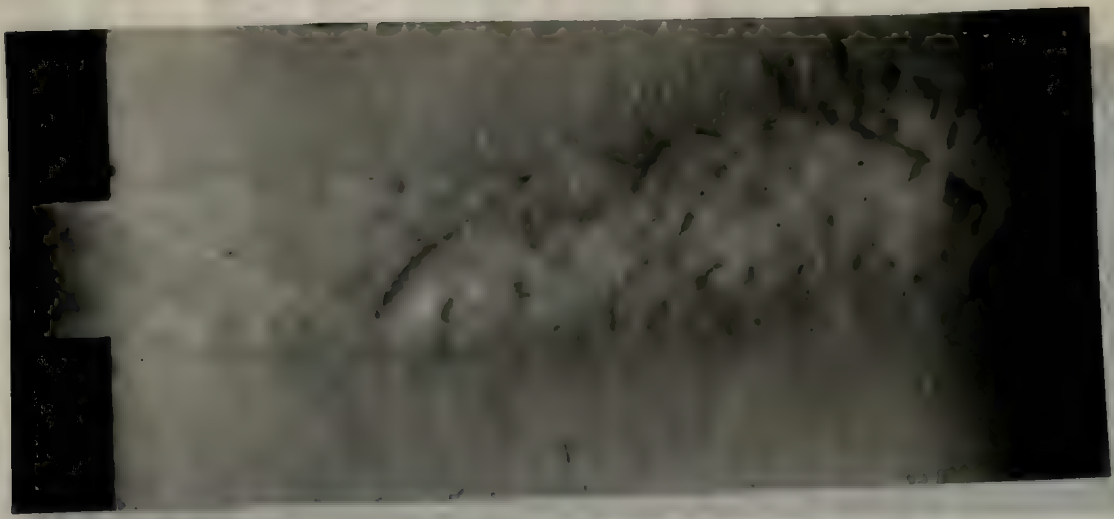


FIGURE V Nozzle #1 $P_e = 95$ $P_2 = 95$ Flash

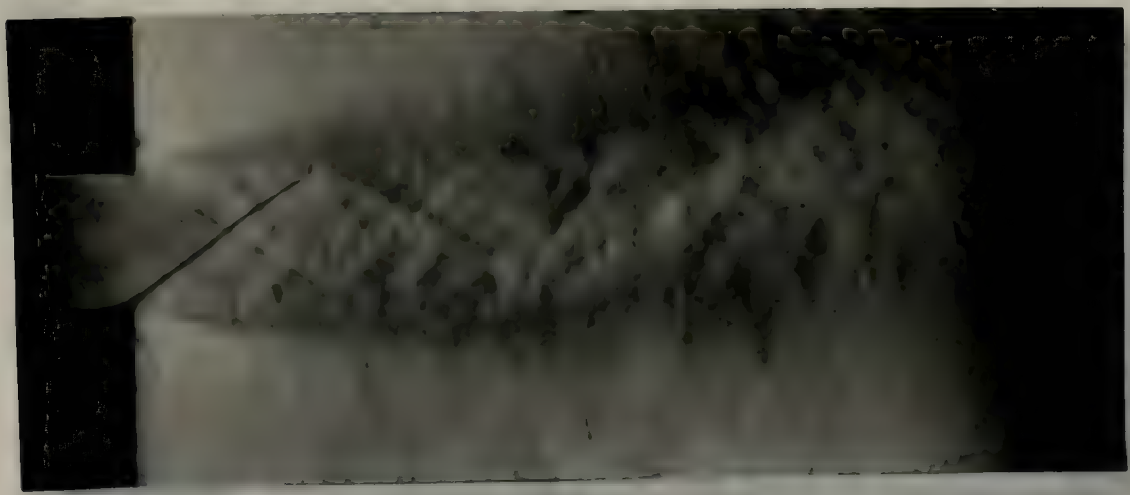


FIGURE VI Nozzle #1 $P_e = 111$ $P_2 = 99$ Flash

FIGURE IV Nozzle #1 $P_e = 74$ $P_2 = 95$ Flash

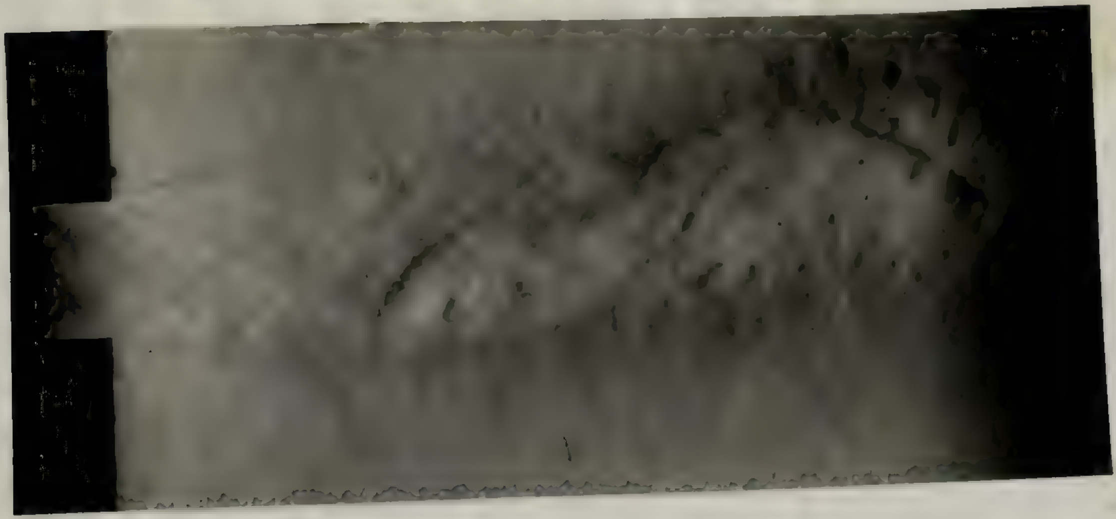


FIGURE V Nozzle #1 $P_e = 95$ $P_2 = 95$ Flash

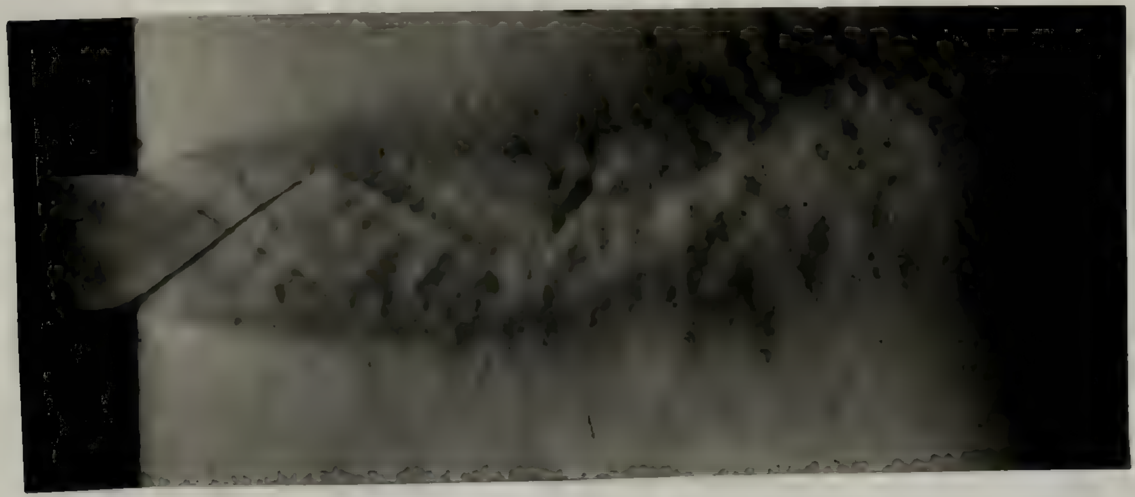


FIGURE VI Nozzle #1 $P_e = 111$ $P_2 = 99$ Flash

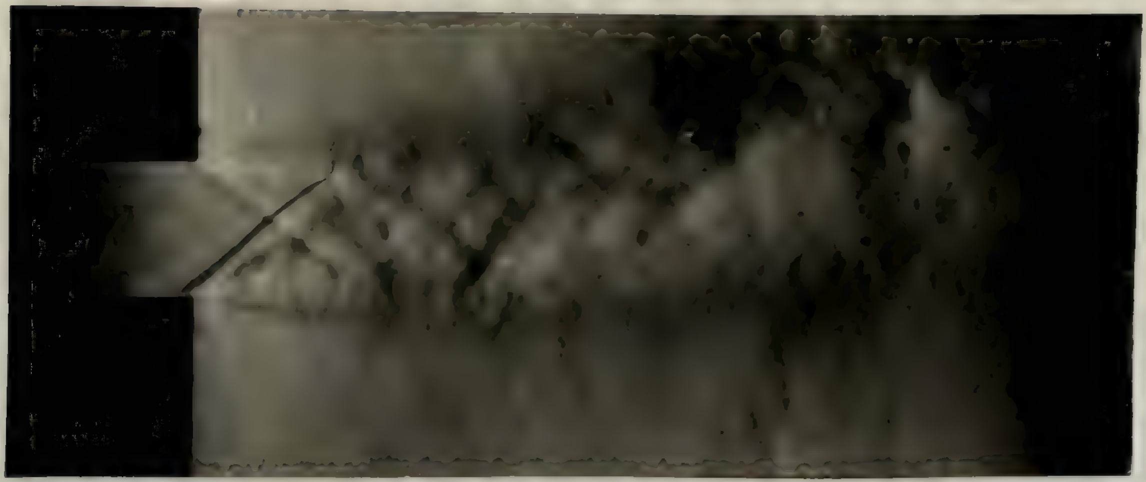


FIGURE VII   Nozzle #1 P_e - 159   Flash
P_2 - 107

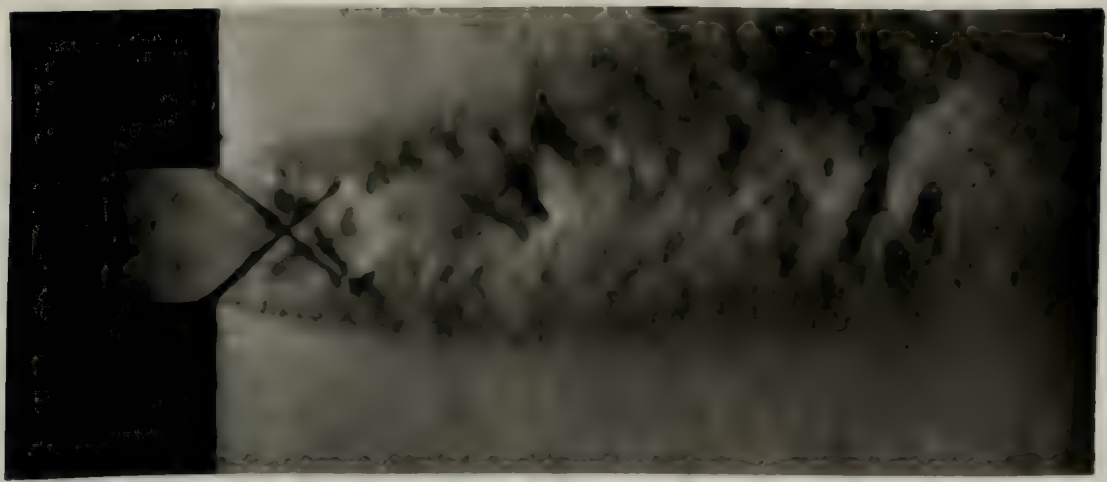


FIGURE VIII   Nozzle #1 P_e - 171   Flash
P_2 - 117

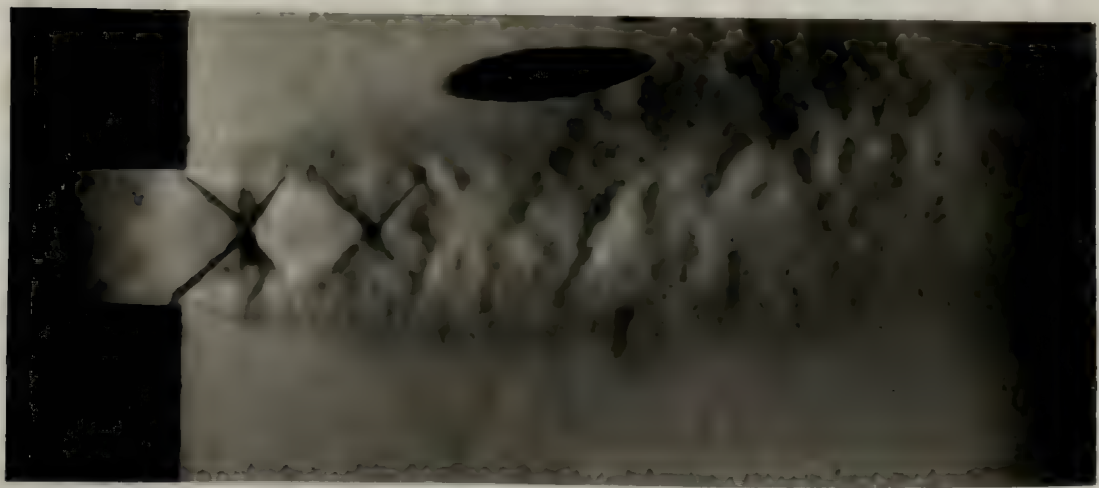


FIGURE IX   Nozzle #1 P_e - 194   Flash
P_2 - 131

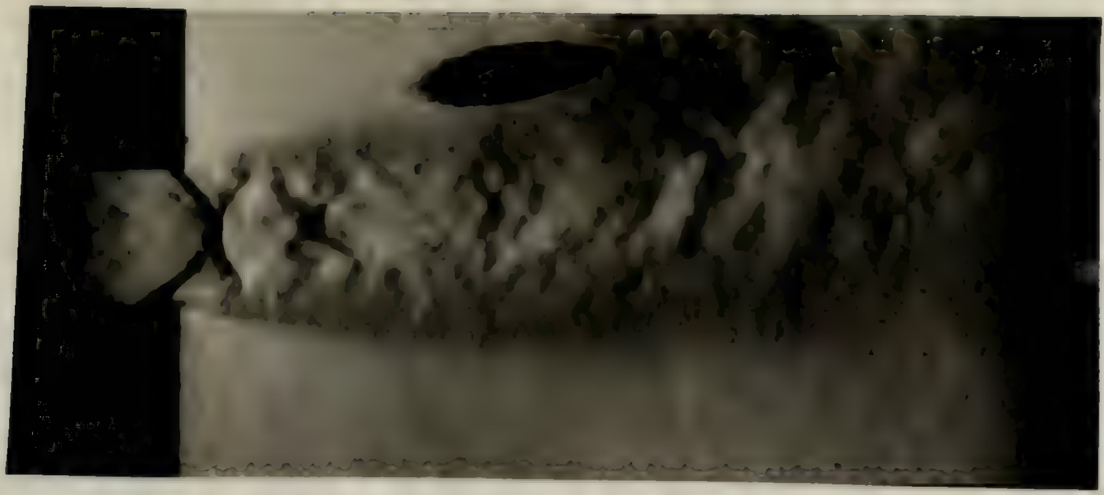


FIGURE X | Nozzle #1 | P_e = 212 P_2 = 210 | Flash

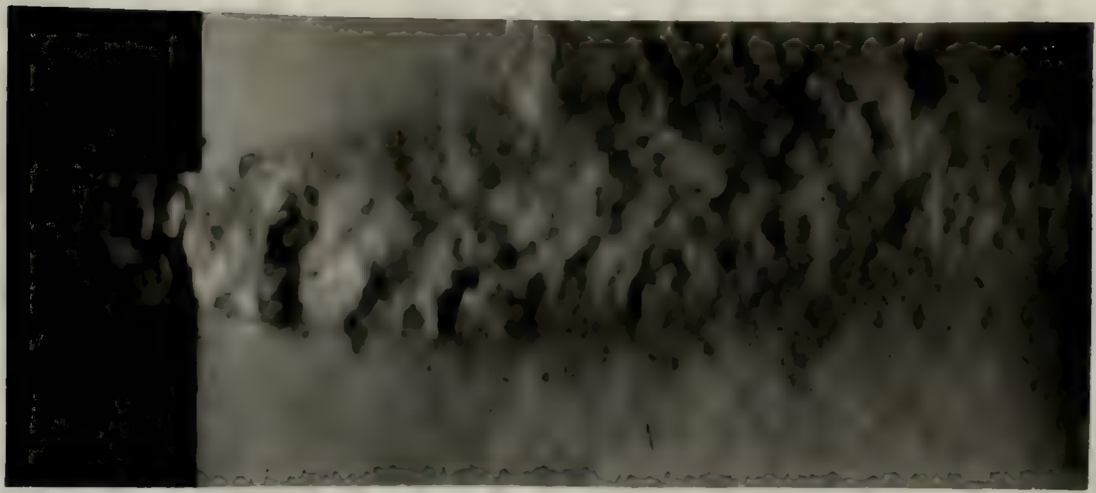


FIGURE XI | Nozzle #1 | P_e = 248 P_2 = 169 | Flash

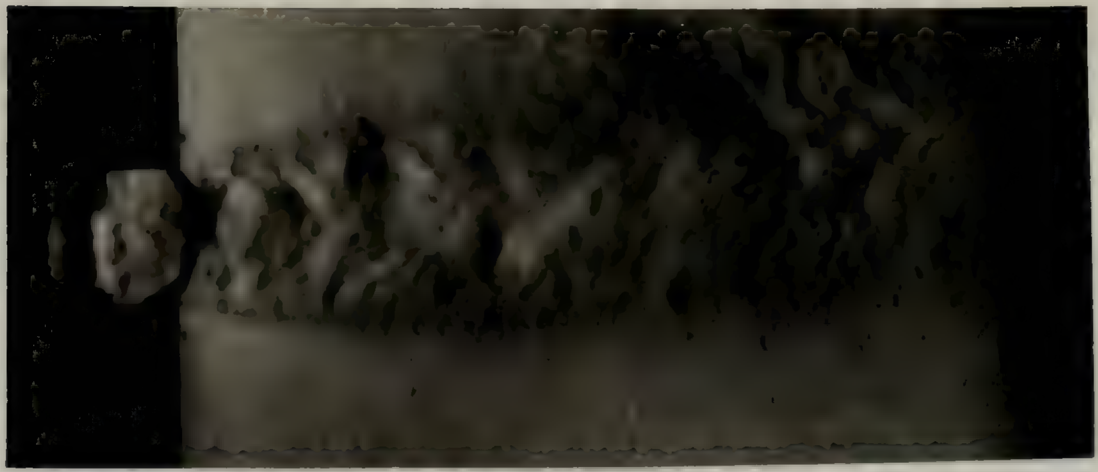
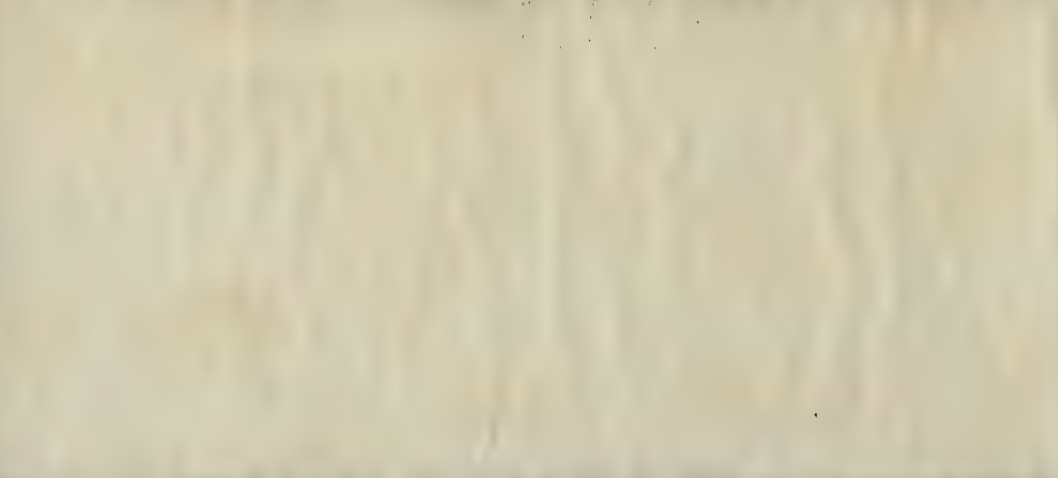
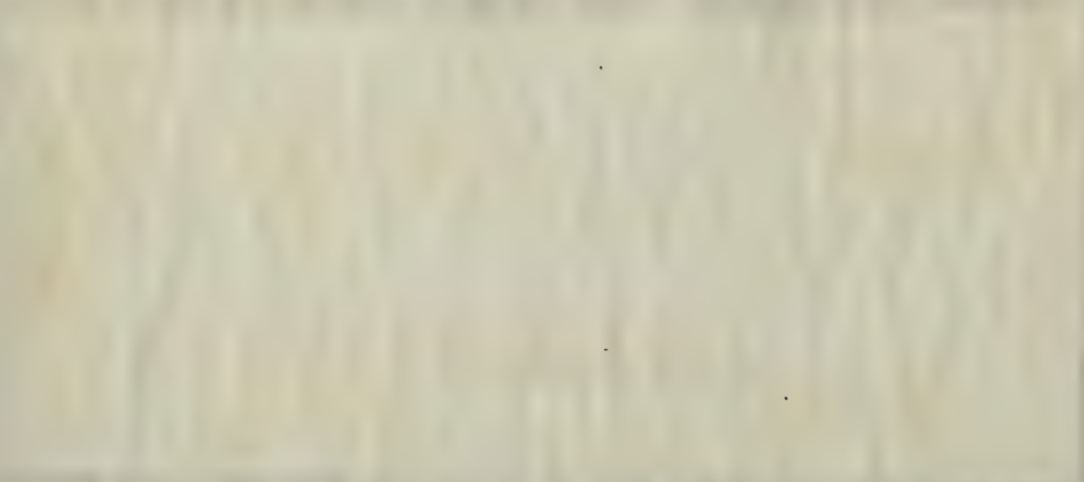
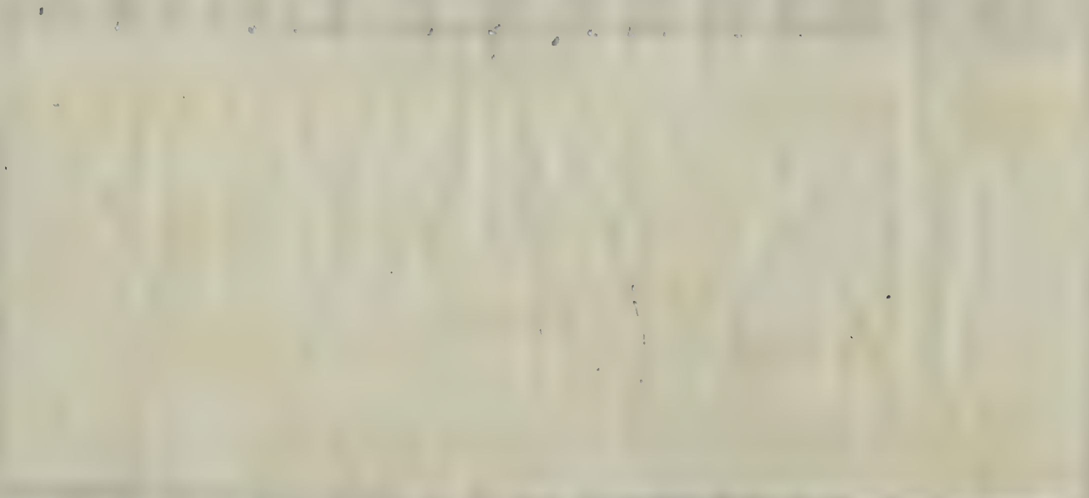


FIGURE XII | Nozzle #1 | P_e = 271 P_2 = 230 | Flash







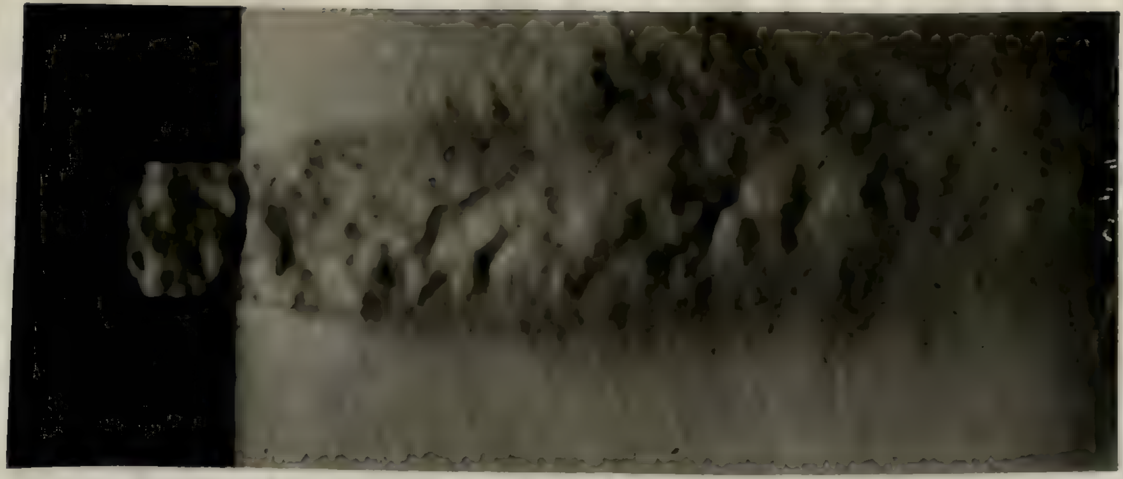


FIGURE XIII | Nozzle #1 | $P_o = 313$
 $P_2 = 288$ | Flash

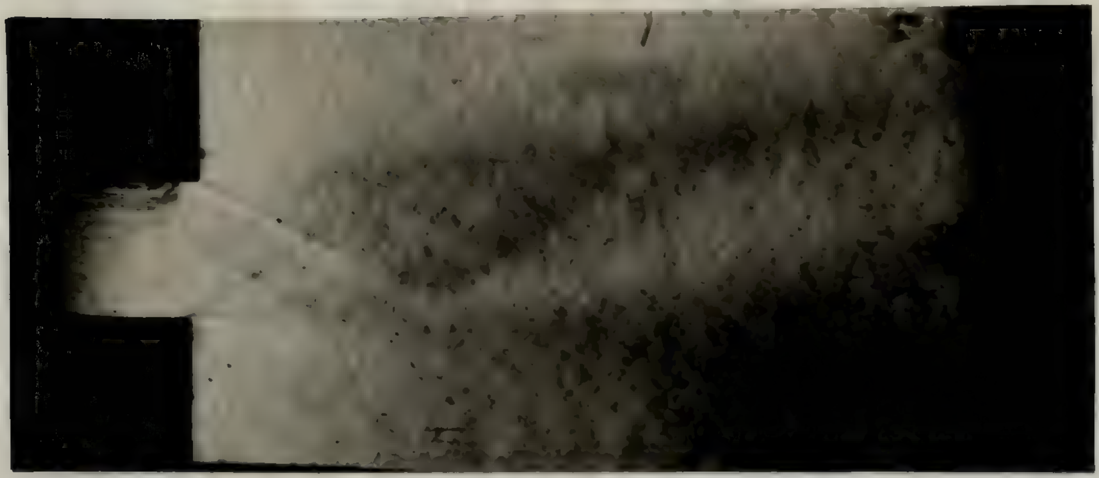


FIGURE XIV | Nozzle #1 | $P_o = 92$
 $P_2 = 95$ | 1/50 Sec.

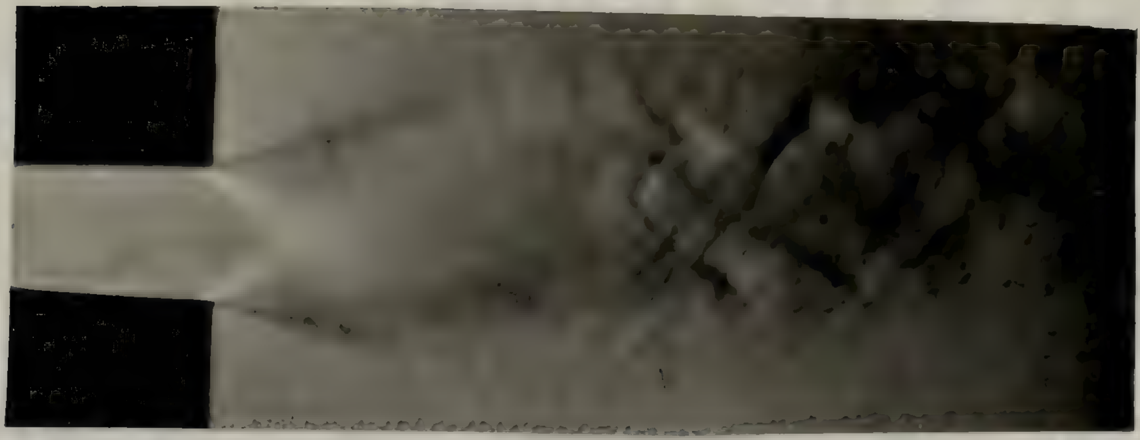
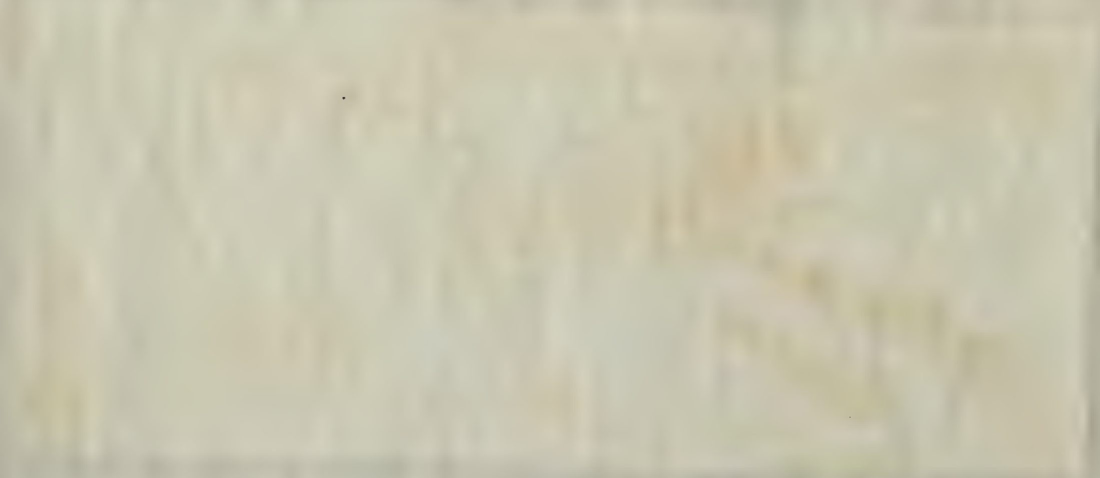
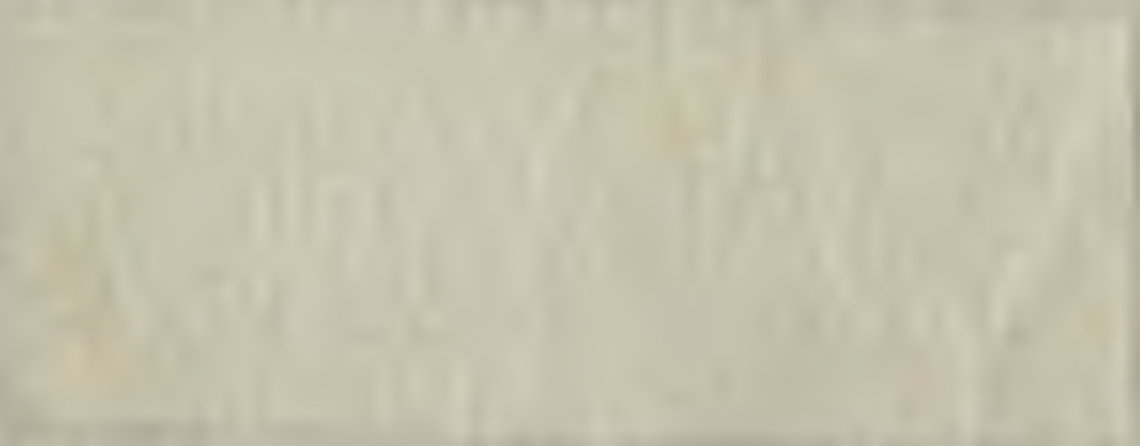


FIGURE XV | Nozzle #2 | $P_o = 73.59$
 $P_2 = 333.5$
 $P_1 = 508.5$ | 1/50 Sec.





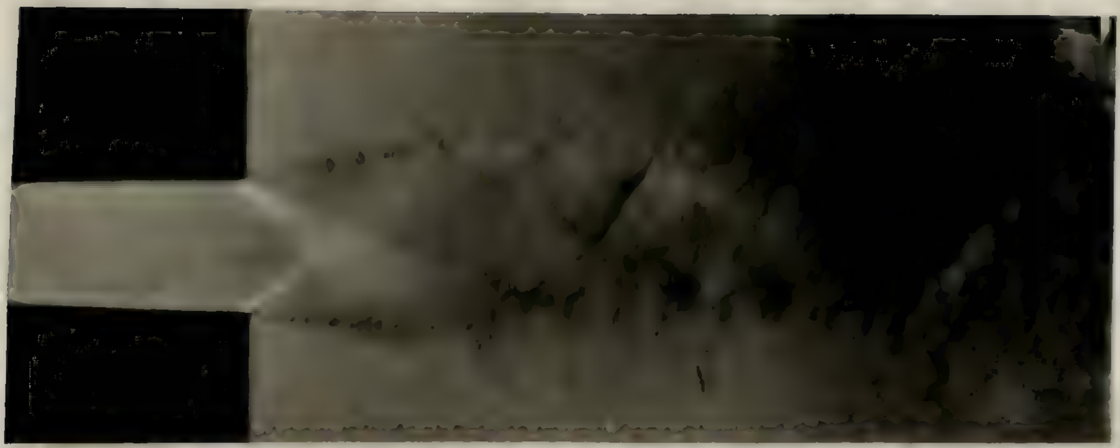


FIGURE XVI | Nozzle #2 | P_e - 94.5, P_2 - 133.5, P_1 - 508.5 | 1/80 Sec.

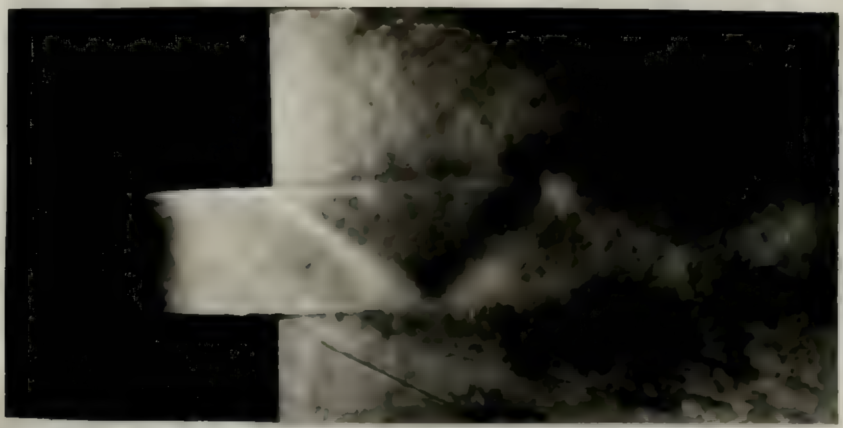


FIGURE XVII | Nozzle #2 | P_e - 132, P_2 - 133, P_1 - 504 | 1/80 Sec.

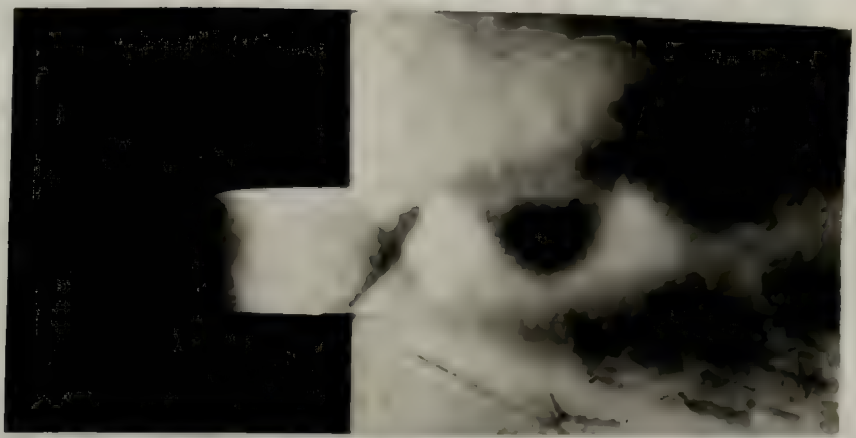


FIGURE XVIII | Nozzle #2 | P_e - 144, P_2 - 138, P_1 - 502 | 1/80 Sec.

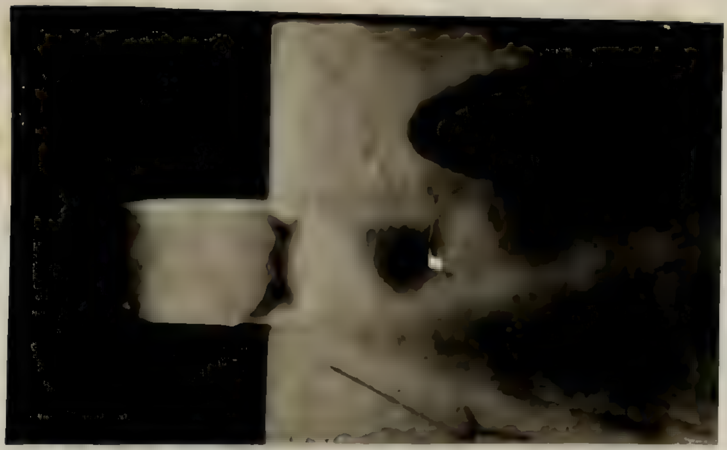


FIGURE XIX Nozzle #2 P_e - 192 1/80 Sec.
P_2 - 155
P_1 - 504

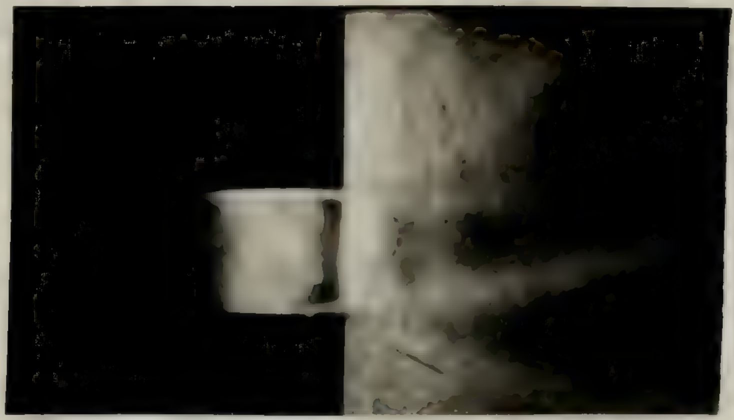


FIGURE XX Nozzle #2 P_e - 217 1/80 Sec.
P_2 - 187
P_1 - 502

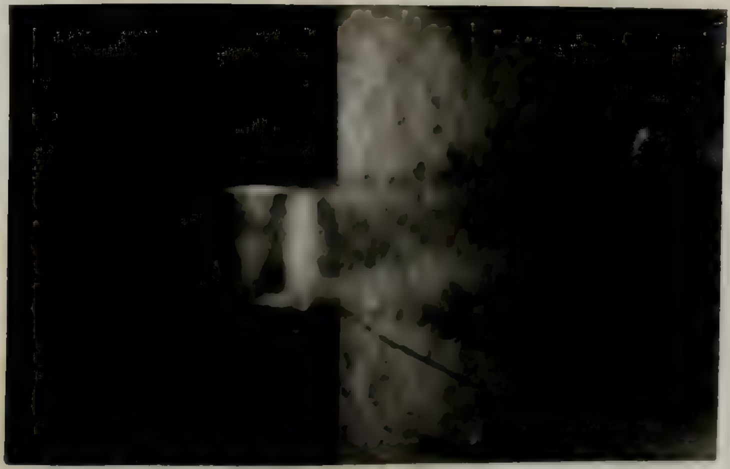


FIGURE XXI Nozzle #2 P_e - 235 1/80 Sec.
P_2 - 217
P_1 - 501

FIGURE XXII Nozzle #2 $P_e - 98$ $P_2 - 130$ $P_1 - 509$ Flash

FIGURE XXIII Nozzle #2 $P_e - 131$ $P_2 - 131$ $P_1 - 508$ Flash

FIGURE XXIV Nozzle #2 $P_e - 156$ $P_2 - 145$ $P_1 - 508$ Flash

FIGURE XXV Nozzle #2 P_e - 146 P_2 - 148 P_1 - 506 Flash

FIGURE XXVI Nozzle #2 P_e - 219 P_2 - 191 P_1 - 504 Flash

FIGURE XXVII Nozzle #2 P_e - 235 P_2 - 215 P_1 - 504 Flash

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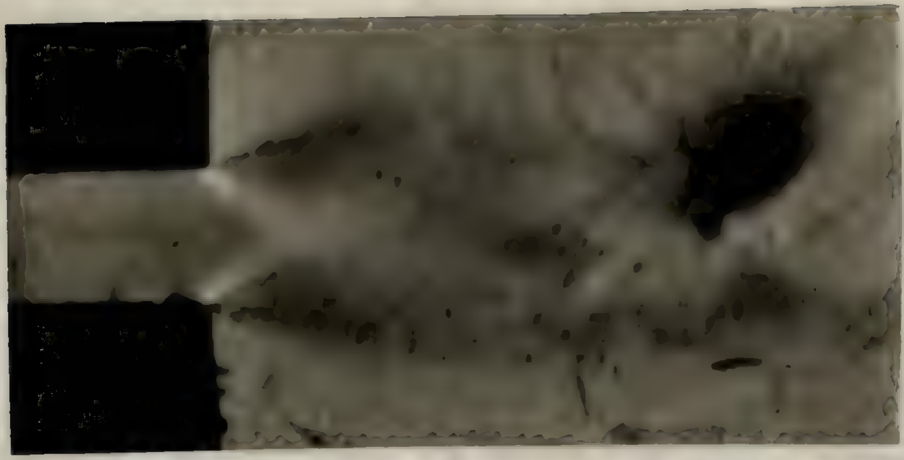


FIGURE XXVIII Nozzle #3 P_e - 90.4 Flash
P_3 - 136.4

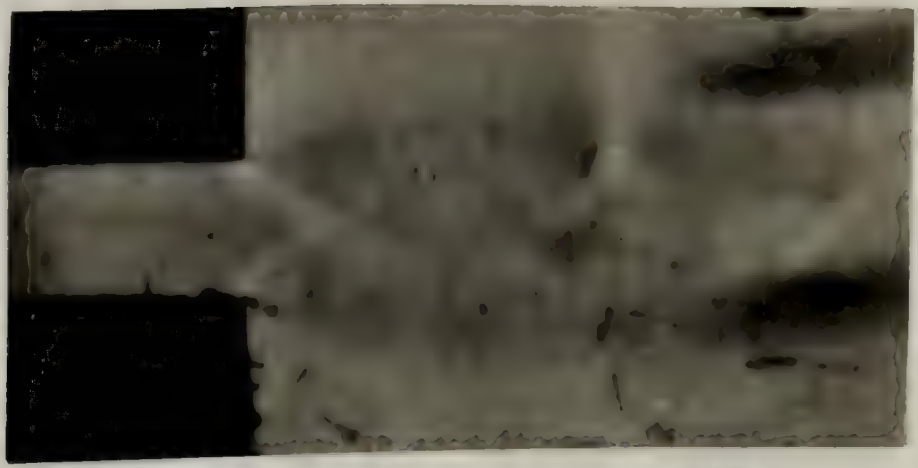


FIGURE XXIX Nozzle #3 P_e - 113.4 Flash
P_3 - 136.4

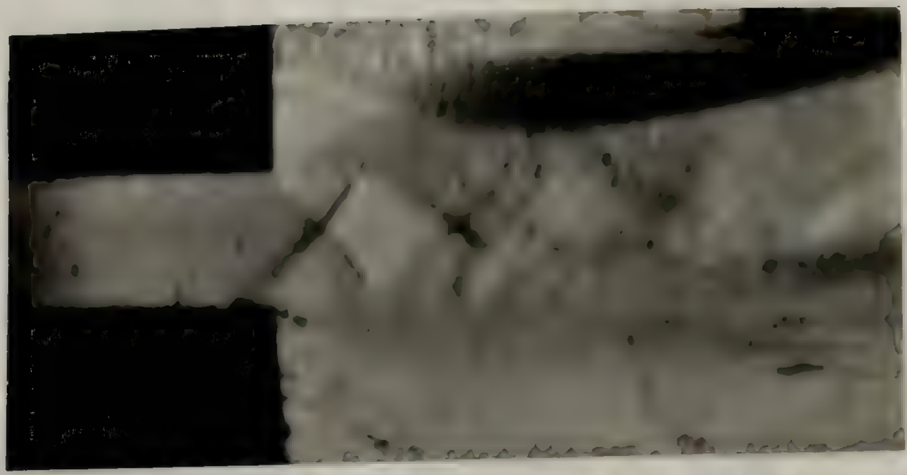


FIGURE XXX Nozzle #3 P_e - 172.4 Flash
P_3 - 136.4

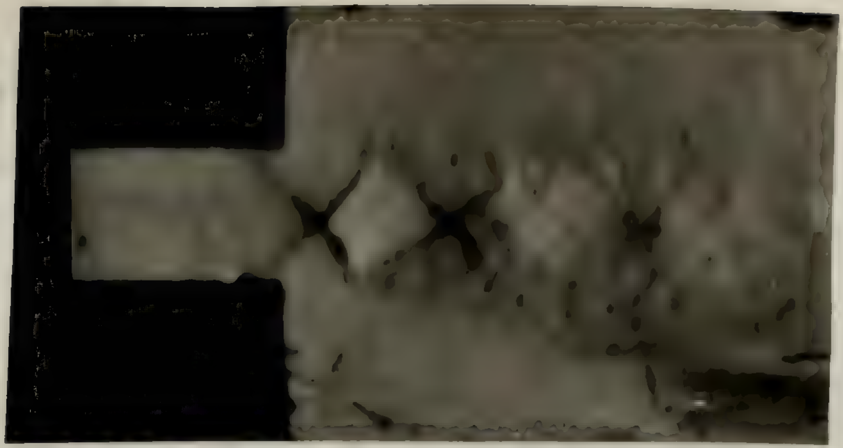


FIGURE XXXI Nozzle #3 P_e - 197.3 Flash
P_3 - 141.4

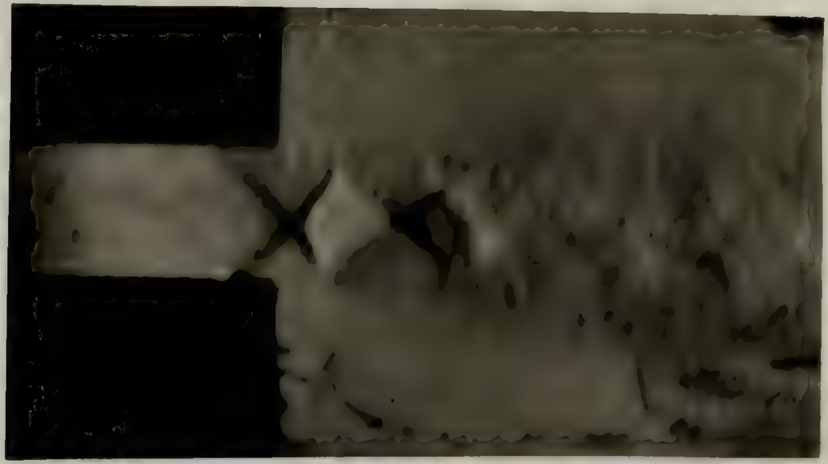


FIGURE XXXII Nozzle #3 P_e - 213.4 Flash
P_3 - 156.4

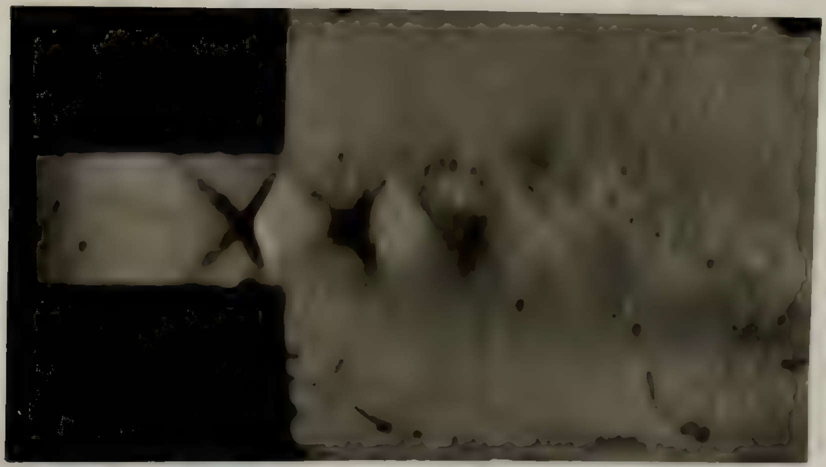


FIGURE XXXIII Nozzle #3 P_e - 238.4 Flash
P_3 - 198.4

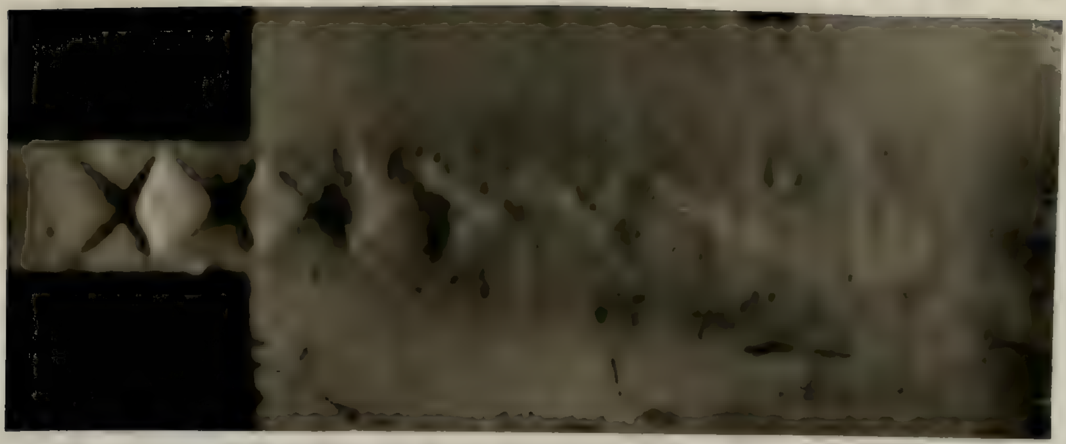


FIGURE XXXIV Nozzle #3 P_e - 261.4 Flash
P_3 - 225.4

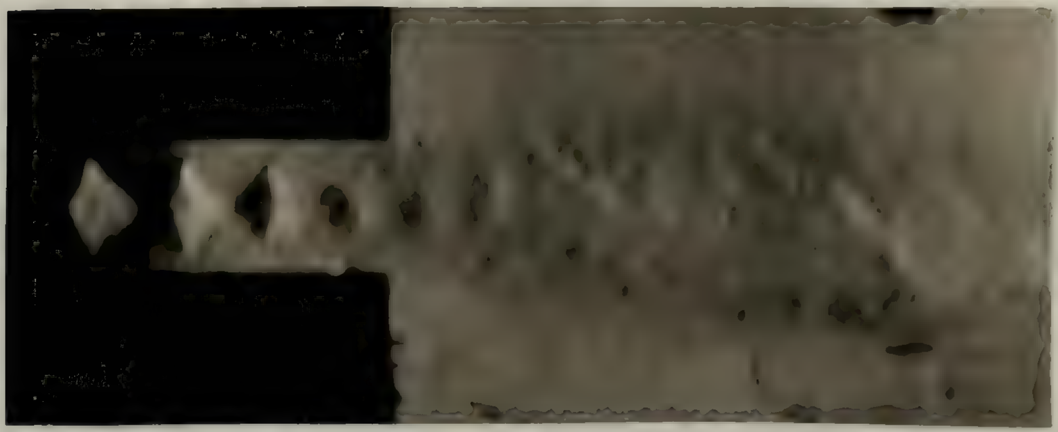


FIGURE XXXV Nozzle #3 P_e - 300.4 Flash
P_3 - 267.4

CONCLUSIONS AND RECOMMENDATIONS

1. Regardless of Mach Number, in supersonic flow the rise of pressure in the exit plane of a practical nozzle is not sudden (in accordance with the theoretical relation) but occurs slowly over a considerable range of exhaust chamber pressure.

2. With thick boundary layer the flow will not support anything resembling a transverse shock.

3. Thickness of boundary layer has the controlling influence on the mechanism by which a supersonic stream adjusts itself to the pressure in the exhaust chamber.

4. It is recommended that further work in this line be carried out with the nozzles mounted parallel to the knife edge of the Schlieren apparatus under the following conditions:

   (a) Use nozzles #1 and #3 of Appendix B.

   (b) Use a nozzle with a tube approximately three (3) inches long at a Mach Number of about 1.39 at exit.

   (c) Use a nozzle with a tube approximately six (6) inches long at a Mach Number of about 1.85 at exit.

5. It is also recommended that investigations of the effect of flow per unit area at the same Mach Number upon the discharge phenomena be carried out.

A P P E N D I X

APPENDIX A -- DETAILS OF PROCEDURE

Reference (3) illustrates that a good shockless nozzle may be designed by the use of the Prandtl Theory; therefore it was decided to use this method as the basis of the nozzle design. The nozzle design was merely a reproduction of the work of Reference (3) but using different area ratios. A theoretical pressure ratio of .10 was chosed for the basic nozzle (Figure XXXVI, Appendix B) with an angle of divergence of 14° 15'. The theoretical Mach number at the exit of this nozzle is 2.152 based on $k = 1.400$. The area ratio is 1.9307. A velocity coefficient of .95 was assumed and the actual Mach number calculated to be 1.85 with a pressure ratio of .124.

It was desired to investigate the effect of Mach number with approximately constant boundary layer thickness on the discharge phenomena. To accomplish this a second nozzle (Figure XXXVII, Appendix B) was designed with an area ratio of 1.287. In order to maintain the same flow per unit area at the exit of the two nozzles the inlet pressure in this nozzle was reduced to two thirds (2/3) of an atmosphere by a specially designed adjustable fitting (Figure XXXIX, Appendix B). With an assumed velocity coefficient of .95, r was calculated as .275 and the Mach number at exit as 1.39. In order to keep both nozzles the same length the angle of divergence was reduced to six (6) degrees. It was believed that any differences that might be caused by this change of angle of divergence would be less than those caused by a change in length which would affect the boundary layer.

In order to observe the effect of boundary layer thickness

APPENDIX A -- DETAILS OF PROCEDURE

Reference (1) illustrates that a good shock-free nozzle may be designed by the use of the Prandtl-Meyer theory therefore it was decided to use this method as the basis of the nozzle design. The nozzle design was merely a reproduction of the work of Reference (1) but using different area ratios. A theoretical pressure ratio of .12 was chosen for the basic nozzle (Figure XXXVI, Appendix D) with an angle of divergence of 14° 10'. The theoretical Mach number at the exit of this nozzle is 2.121 based on k = 1.400. The area ratio is 1.9397. A velocity coefficient of .95 was assumed and the actual Mach number calculated to be 1.85 with a pressure ratio of .124.

It was desired to investigate the effect of Mach number with approximately constant boundary layer thickness on the discharge phenomena. To accomplish this a second nozzle (Figure XXXVII, Appendix D) was designed with an area ratio of 1.287. In order to maintain the same flow per unit area at the exit of the two nozzles the inlet pressure to this nozzle was reduced to two thirds (2/3) of an atmosphere by a specially designed adjustable fitting (Figure XXXIX, Appendix D). With an assumed velocity coefficient of .95, r was calculated as .275 and the Mach number at exit as 1.55. In order to keep both nozzles the same length the angle of divergence was reduced to six (6) degrees. It was believed that any differences that might be caused by this change of angle of divergence would be less than those caused by a change in length which would affect the boundary layer.

In order to observe the effect of boundary layer thickness

on the discharge phenomena for the same Mach number at exit, it was decided to add to the basic nozzle (Nozzle 1) a straight constant area section of such length as to reduce the Mach number of Nozzle #1 (1.85) to the Mach number of Nozzle #2 (1.39). To eliminate the possibility of shock formation at the junction of the nozzle and tube it was decided to fabricate another nozzle with the straight portion integral with the nozzle itself (Figure XXXVIII). By use of data obtained from Reference (6) the length of tube necessary was calculated to be 10.35 inches. This figure was regarded as highly approximate due to the use of a two dimensional tube instead of the circular section upon which the data of Reference (6) is based.

Provision was made for pressure measurement by mercury manometer at a point one eighth (1/8) inch from the nozzle or tube exit and in the discharge chamber of all nozzles by a .020 inch diameter hole in the steel contour.

All pictures were taken with the axis of the nozzle perpendicular to the knife edge of the Schlieren apparatus described adequately in Reference (1).

The pictures designated "Flash" were made by using the Edgerton Flash Unit also described in Reference (1). This gave an exposure time of approximately $.5 \times 10^{-6}$ seconds. A few pictures were taken using a steady light source and an exposure time of 1/80 seconds, to show the difference in detail of pictures obtained by the use of the two different methods.

APPENDIX B---EXPERIMENTAL DATA

TABLE I

PRESSURE READINGS, NOZZLE I 19 JULY, 1946

P_e = Exhaust Chamber Pressure, mm. Hg.

P_2 = Pressure in Exit of Nozzle, mm. Hg.

P_a = Nozzle Inlet Pressure, Atmospheric

T_1 = Inlet Temperature, Degrees F.

P_e	P_2	P_a	T_1
74	95	761.2	85
95	95		
116	99		
133	104		
153	109		
169	116		
181	122		
196	131		
220	136		
241	197		
260	228		
287	230		
324	289		
370	341		
424	407		
473	464		
527	520		

TABLE XI

PRESSURE READINGS, NOZZLE # 2 13 JULY, 1946

P_e = Exhaust Chamber Pressure, mm. Hg.

P_2 = Pressure in Exit of Nozzle, mm. Hg.

P_1 = Nozzle Inlet Pressure, mm. Hg.

P_a = Atmospheric Pressure, mm. Hg.

T_1 = Nozzle Inlet Temperature, Degrees F.

P_e	P_2	P_1	P_a	T_1
71	133	502	761.5	85
86	133	502		
94	133	502		
113	133	502		
131	133	502		
144	138	502		
151	143	502		
171	146	502		
183	150	502		
192	155	502		
211	178	502		
217	188	502		
244	227	502		
296	288	502		
340	338	502		
418	417	502		

TABLE III

PRESSURE READINGS, NOZZLE #3 — 22 JULY, 1946

P_e = Exhaust Chamber Pressure, mm. Hg.

P_3 = Pressure at Exit of Tube, mm. Hg.

P_2 = Pressure at Tube Inlet (Nozzle Exit), mm. Hg.

P_a = Nozzle Inlet Pressure, Atmospheric

T_1 = Inlet Temperature, Degrees F.

P_e	P_3	P_2	P_a	T_1
80.4	135.4	93.4	764.4	85
97	135			
113	135			
129	135			
140	135			
158	135			
185	137			
197	141			
206	148			
213	156			
228	181			
238	198			
246	213			
261	225			
288	255			
300	267			
358	229			

TABLE IV

NOZZLE CHARACTERISTICS

A_t = Cross-sectional Area at Throat

A_e = Cross-sectional Area at Exit

r_t = Theoretical Ratio of Exit Pressure to Inlet Pressure

r_a = Actual Ratio of Exit Pressure to Inlet Pressure

M_t = Theoretical Mach Number at Exit (Frictionless Flow)

M_a = Actual Mach Number at Exit

C_v = Assumed Velocity Coefficient

w = Flow in Pounds per Second

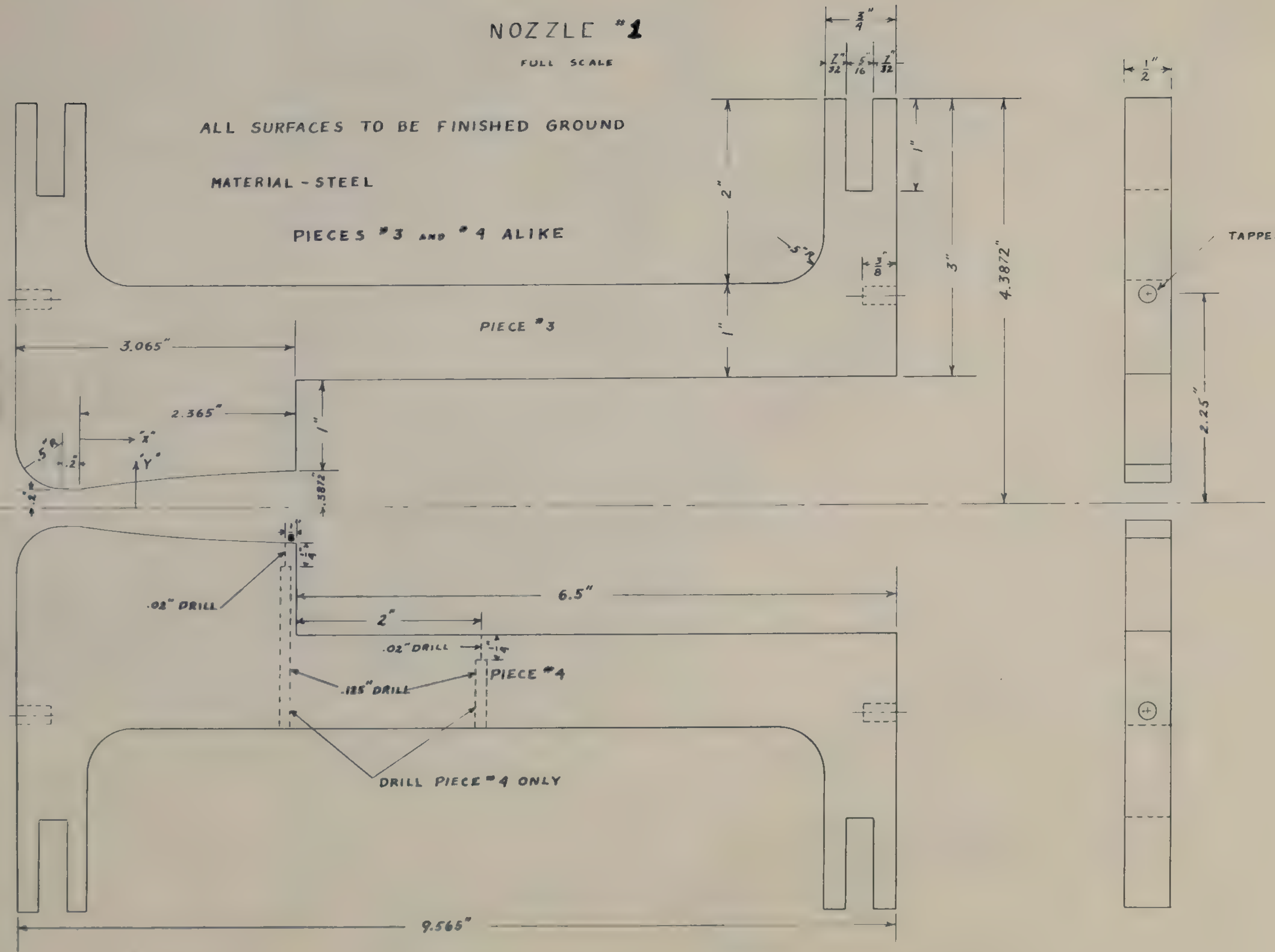
G = Flow per Unit Area at Exit, Pounds per Squarefoot per Second

T_1 = Inlet Temperature, Degrees F.

P_1 = Inlet Pressure, Atmospheres

NOTATION	NOZZLE # 1	NOZZLE # 2	NOZZLE # 3
A_t/A_e	1.9307	1.2870	1.9307
r_t	.1000	.2200	.1000
r_a	.1240	.2750	.1833
M_t	2.1520	1.6180	2.1520
M_a	1.8500	1.3900	1.3900
C_v	.95	.95	
P_1	1.0000	.6667	1.0000
T_1	85	85	85
w	.0676	.0676	.0676
G	25.2000	25.2000	25.2000

FIGURE XXXVI



OFFSETS NOZZLE #1

"X"	"Y"	
0	.2000	STRAIGHT SLOPI
.87	.3100	
.95	.3200	CURVED PORT
1.00	.3250	
1.10	.3360	
1.20	.3470	
1.30	.3550	
1.40	.3635	
1.50	.3640	
1.60	.3740	
1.70	.3790	
1.80	.3830	
1.90	.3850	
2.00	.3860	
2.10	.3865	
2.20	.3870	
2.24	.3872	ST. HORZ. PORTIO
2.365	.3872	

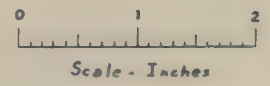


FIGURE XLVII

OFFSETS NOZZLE "2"

Portion	"X"	"Y"
STRAIGHT SLOPING	0	.3000
	1.00	.3500
	1.10	.3550
	1.20	.3600
CURVED PORTION	1.30	.3640
	1.40	.3675
	1.50	.3710
	1.60	.3742
	1.70	.3775
	1.80	.3800
	1.90	.3825
	2.00	.3850
	2.10	.3860
	2.20	.3870
ST HORZ PORTION	2.24	.3872
	2.305	.3872

NOZZLE #2
FULL SCALE

ALL SURFACES TO BE FINISHED GROUND

MATERIAL - STEEL

PIECES #5 AND #6 ALIKE

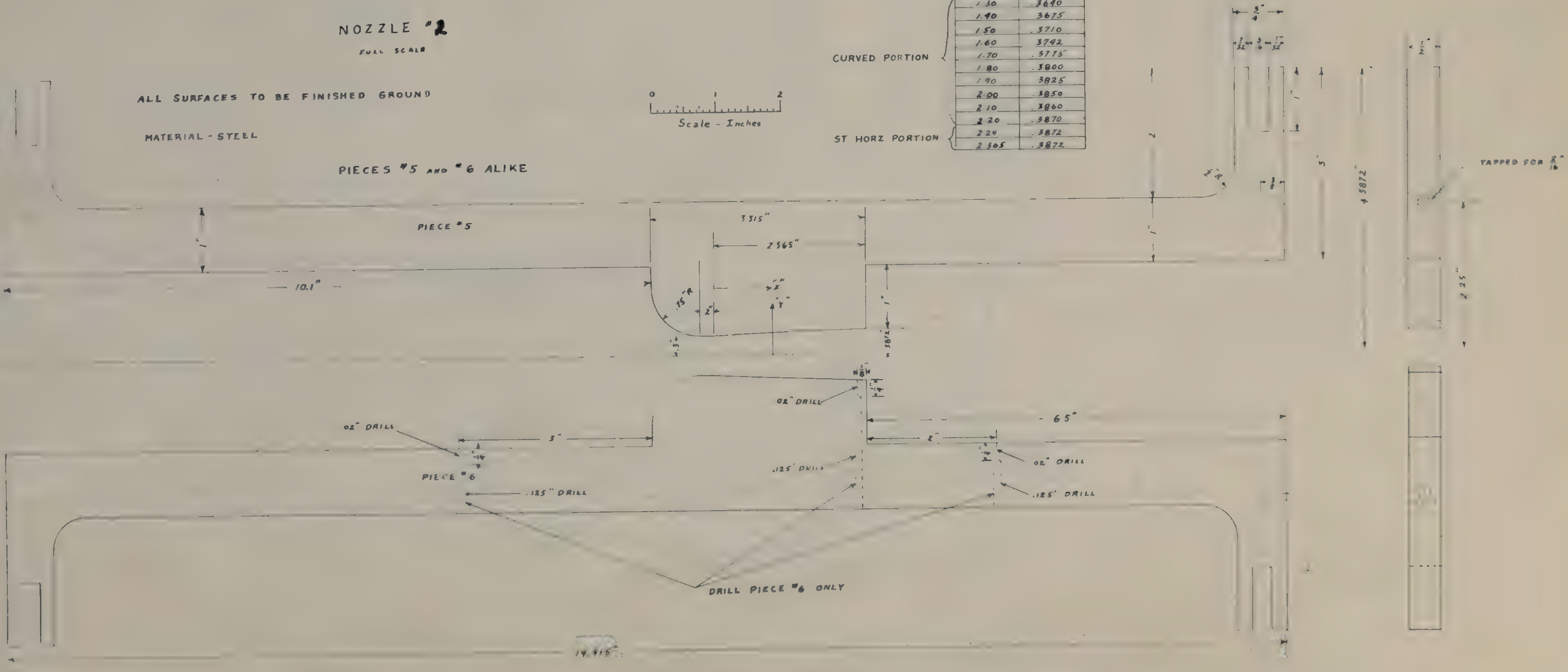


FIGURE XXXVIII

OFFSETS NOZZLE #3

"X"	"Y"	
0	.2000	STRAIGHT SLOPING
87	.3100	
95	.3200	CURVED PORTION
1.00	.3250	
1.10	.3360	
1.20	.3470	
1.30	.3550	
1.40	.3635	
1.50	.3690	
1.60	.3740	
1.70	.3790	
1.80	.3830	
1.90	.3850	
2.00	.3860	
2.10	.3865	
2.20	.3870	
2.24	.3872	ST. HORZ. PORTION
2.365	.3872	

NOZZLE #3

FULL SCALE

ALL SURFACES TO BE FINISHED GROUND

MATERIAL - STEEL

PIECES #1 AND #2 ALIKE

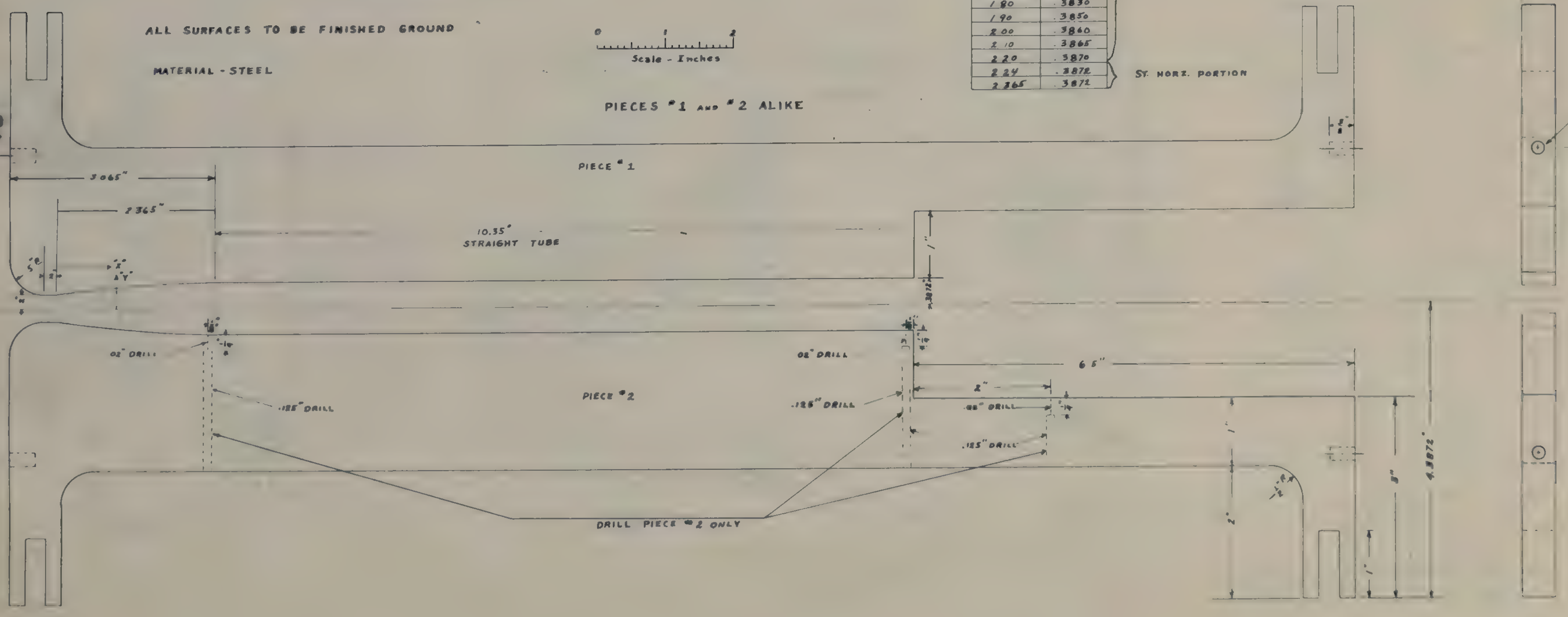
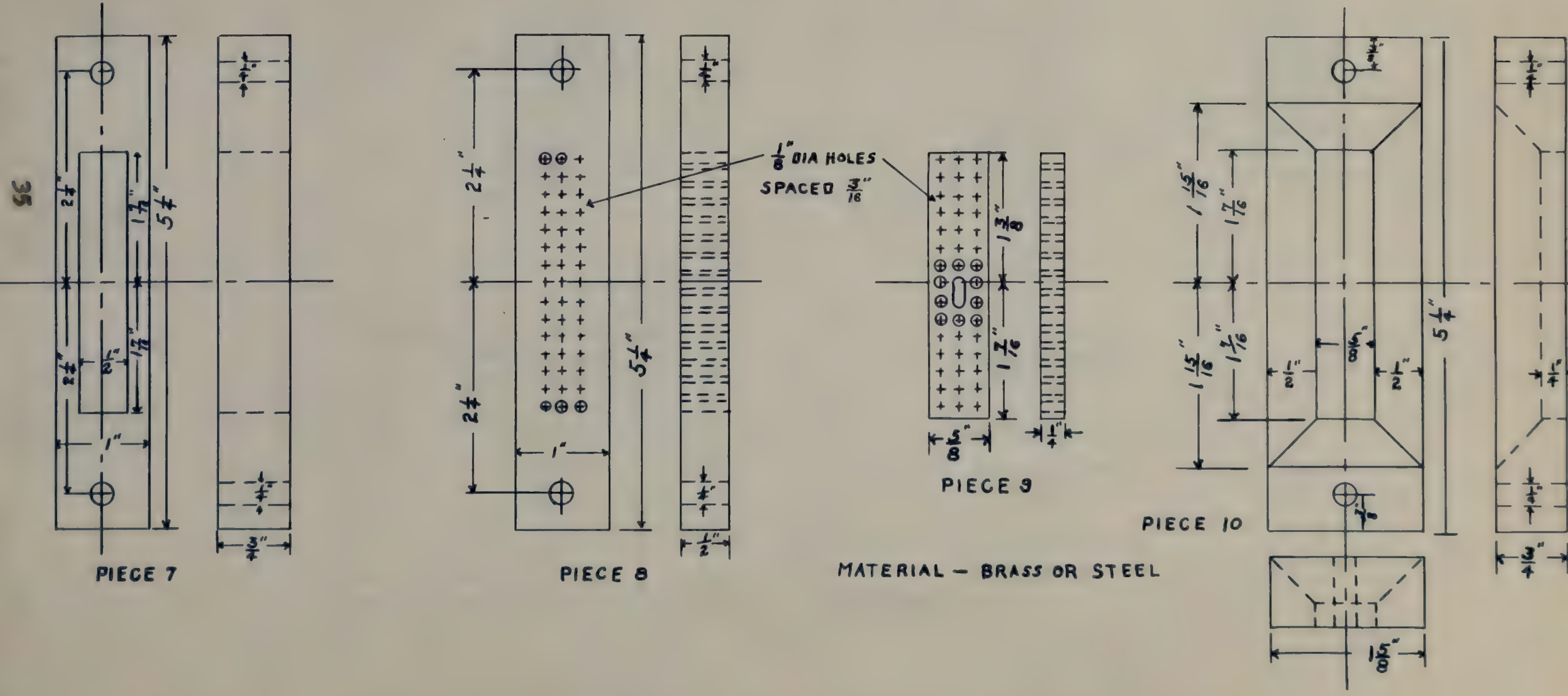


FIGURE XXXIX

REDUCING FITTING



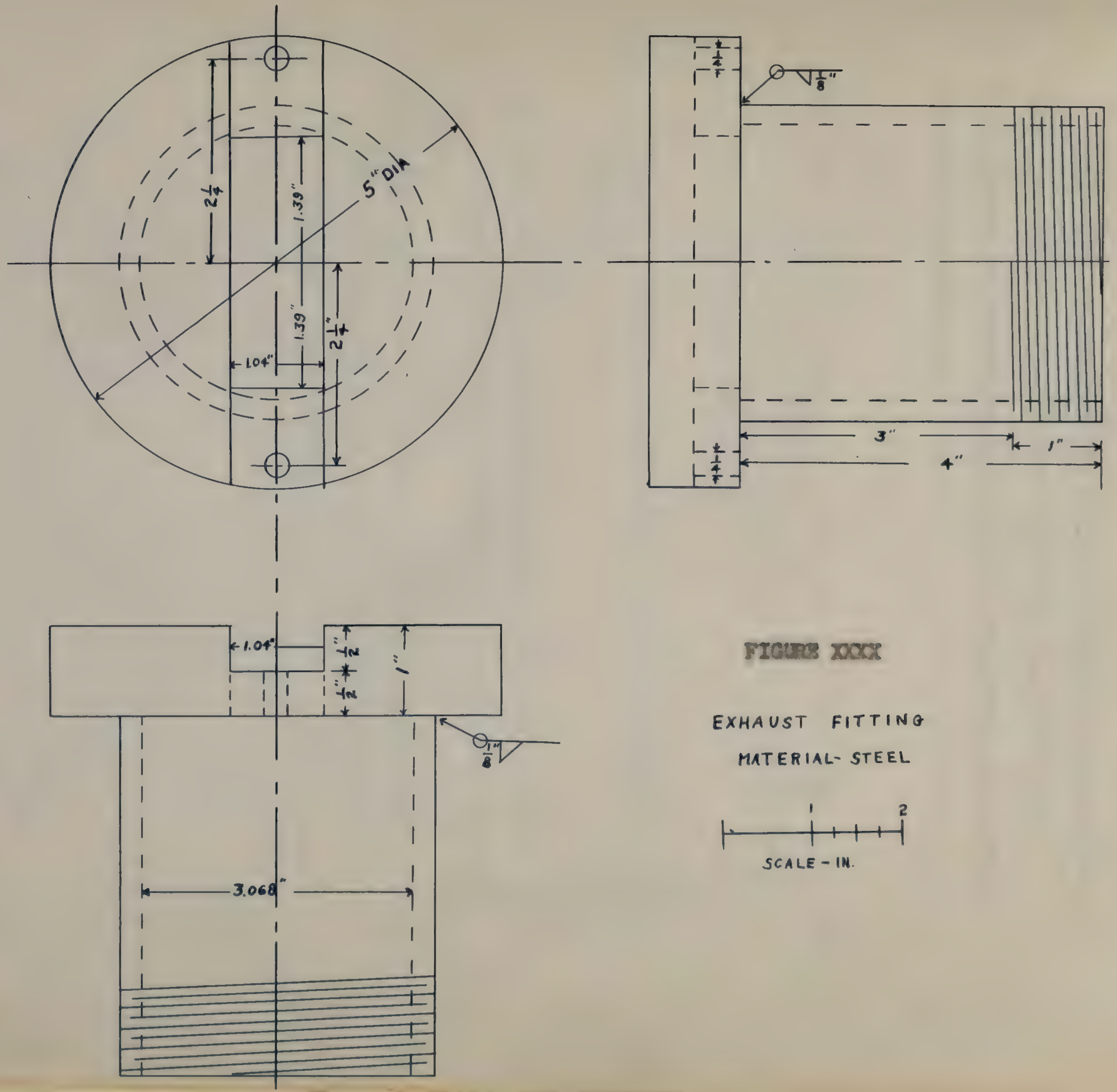


FIGURE XXXX

EXHAUST FITTING

MATERIAL- STEEL

SCALE - IN.

APPENDIX C -- LOCATION OF ORIGINAL DATA

All of the original nozzle design calculations, photographic negatives, and the nozzle profiles are in the possession of Mr. E. P. Neumann of the Mechanical Engineering Department, Massachusetts Institute of Technology.

APPENDIX D — BIBLIOGRAPHY

1. L.A. DeFrate, "Investigation of Supersonic Flow in Nozzles and Tubes by the Schlieren Method", Master's Thesis, M.I.T., 1943.

2. Lieut. F.M. Huron, U.S.N., and Lieut. W.R. Nelson, U.S.N., "Investigation of Supersonic Flow in Nozzles and Tubes", Master's Thesis, M.I.T., 1944.

3. Lt. Comdr. J.W. Dolan, Jr., U.S.N. and Lt. Comdr. R.V.Laney, U.S.N., "Experimental Study of Supersonic Flow in the Prandtl Nozzle", Master's Thesis, M.I.T., 1944.

4. Joseph H. Keenan, Thermodynamics, John Wiley and Sons, New York, 1941.

5. Dr. A. Stodola, Steam and Gas Turbines, Vol. II, translated by Dr. Louis C. Lowenstein, McGraw Hill Book Co., Inc., New York.

6. J.H. Keenan and E.P. Neumann, "Friction in Pipes at Supersonic and Subsonic Velocities", National Advisory Committee for Aeronautics, Technical Note # 963.

[illegible]

1. [illegible] [illegible] and Tubes by the Schlieren Method", Master's Thesis, M.I.T., [illegible].

2. [illegible] [illegible] Investigation of Supersonic Flow in Nozzles and Tubes", Master's Thesis, M.I.T., 1944.

3. [illegible]

4. [illegible]

5. [illegible]

6. [illegible]